

RESIDUAL STRENGTH EVALUATION OF A TRANSPORT AIRCRAFT FUSELAGE STIFFENED PANEL

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ABSTRACT

Wide-bodied transport aircraft are susceptible to the problem of fatigue cracking of the pressurized cabin. In order to design against catastrophic failure due to such fatigue cracking, the residual strength of a transport aircraft fuselage stiffened panel is analyzed for a crack spanning over two bays. A finite element model of the entire fuselage structure of a transport aircraft was modeled and a linear elastic stress analysis was carried out to identify the location of maximum tensile stress location susceptible for initiation of a fatigue crack. This is followed by fracture mechanics analysis of a growing crack in the panel to assess its crack arrest capability. Residual strength assessment was carried out for the panel and it was observed that residual strength of stiffened panel decreases with increase in the crack length, As the crack length approaches stiffener location the stiffener will share the maximum load, resulting in higher residual strength in the skin leading to crack arrest. Also, the residual strengths of the skin, stiffener and the rivets are estimated individually to generate panel strength diagram, which demonstrates the crack arresting capability of the stiffened panel.

Keywords: Fatigue cracking, Residual strength, Crack arrest, Panel strength diagram

1.0 INTRODUCTION

Structural failures have frequently been observed in practice even when applied load suggests that stress levels should be significantly lower than yield stress. These premature failures are often attributed to defects in the structure that give rise to a local elevation of the stresses. These defects are in the form of sharp flaws, cracks or some other form of damage. This results in stress elevation in the vicinity of the defect, possibility of local material failure, and may give rise to a progressive enlargement of damage region such as through a crack growth mechanism. It is vital to establish whether a damage in a structure have an impact on its integrity. Over the last 50 years, fracture mechanics concepts have been introduced and validated, with which the engineers and scientists of today are now well positioned to make accurate judgment on the integrity of flawed structures.

The prevention of crack initiation in structures is often extremely difficult task, as a result many industries can tolerate a small amount of crack growth in structures. This is permissible provided it does not seriously impact the integrity and the operation of the structure and hence progressive fracture in a structure is continuously monitored to ensure that the crack length does not become critical and lead to catastrophic failure. From the designer point of view, it is necessary to design structures to withstand effects of damage in addition to requirement such as strength, stiffness, etc.

The current industry practice requires that a 2-bay crack must get arrested under a specified design limit load. Figure1 shows a longitudinal crack in the pressurized skin of a transport aircraft. Such a crack can occur due to the impact of a broken fragment of an uncontained engine failure. The crack configuration considered is as follows. The central bulkhead is broken and the skin is cracked on the top of this bulkhead. Due to pressure loading such a crack can meet the unstable fracture condition. In such a case, the fast fracturing crack must get arrested by the

two outer bulkheads (under the action of the Design Limit Load). The two bay crack problem is depicted in Figure 1.

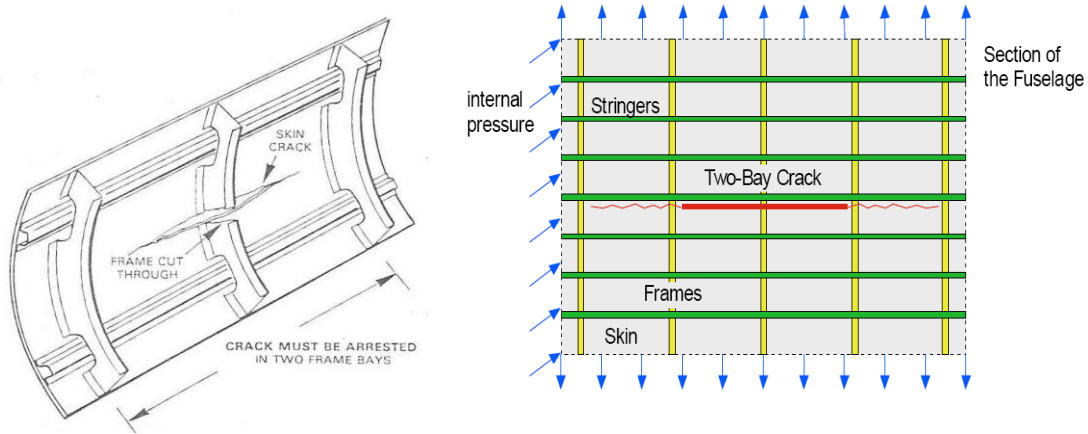


Figure 1. Schematic of a two bay crack problem

Both analytical and numerical methods can be used to study the 2-bay crack problem and generate panel strength diagrams for a damage tolerance evaluation. Displacement compatibility method is an analytical approach. Rivet flexibility, stringer bending are some of the important aspects that can be incorporated in the analysis. On the other hand finite element modeling can handle a host of other complexities like load eccentricity, material non-linearity and cracked outer stiffeners.

2.0 FINITE ELEMENT ANALYSIS

2.1 Fuselage

In the present study fuselage structure of a transport aircraft SARAS is modeled and analyzed for pressurization load case for the identification of the maximum tensile stress location in the top skin of the fuselage and thereby identifying the critical bay.

2.1.1 FE modeling

It involves modeling the different components of the fuselage such as bulkheads, stringers skin, main doors, exit doors with windshield frame, wing and fuselage attachment brackets etc. The solid geometry of all the fuselage components in the form of CATIA models was used for the finite element modeling using MSC Patran. The modeled fuselage is a 2D Integral FE model and there are totally around 114821 elements comprising of QUAD4, TRIA3, 1D beam elements and RBE2 elements. Various material properties for the different components that are considered in the study are shown in Table 1.

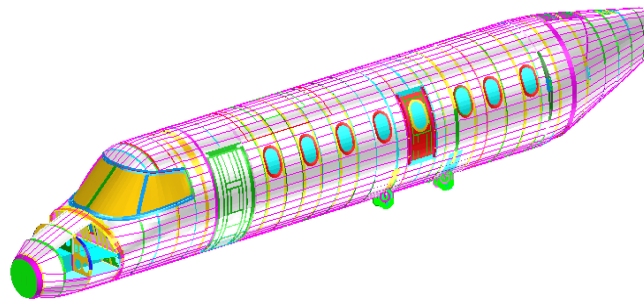


Figure 2. FE Model of Complete fuselage

2.1.2 Material Properties

Table 1: Material properties of the fuselage components

Component	Material	Properties
Skin Bulkhead Stringer Wind shield frames Rivets Connecting pins	Al 2024-T3	Modulus of Elasticity, $E = 70.6\text{GPa}$ Poisson's ratio $\nu = 0.3$ Ultimate tensile strength $\sigma_{ult} = 414\text{ MPa}$ Fracture toughness, $K_{IC} = 35\text{ MPa}\sqrt{\text{m}}$
Attachment bracket connecting bolts	Mild steel	Modulus of Elasticity, $E = 20.6\text{ GPa}$ Poisson's ratio $\nu = 0.28$
Window& Windshield Glass	Perspex glass	Modulus of Elasticity, $E = 7.166\text{ GPa}$ Poisson's ratio $\nu = 0.22$

2.1.3 Loads and boundary conditions

The fuselage was constrained at the wing fuselage attachment brackets in all DOF $U_x, U_y, U_z, R_x, R_y, R_z = 0$) and the link to attachment bracket connecting pin is allowed to rotate about its axis. The fuselage was subjected to an internal pressure of 6.9 Psi (0.048069 MPa) normal to the surface of the fuselage components between the two pressure bulkheads.

2.1.4 FE analysis of fuselage

A linear static analysis was conducted under pressurization loads to evaluate the maximum tensile stress location and the structural response. The maximum tensile stress location at the top skin of the fuselage was identified and the stiffened panel dimensional configuration was marked for the crack arrest capability demonstration

2.2 Stiffened panel

The stiffened panel selected from the global analysis of fuselage is of 1760 mm in width and 1260 mm in height and there are five bulkhead stiffeners having a spacing of 360 mm and 400mm alternately, as shown in the Figure 3 and there are totally nine stringers along the longitudinal direction spaced at a distance of 140 mm.

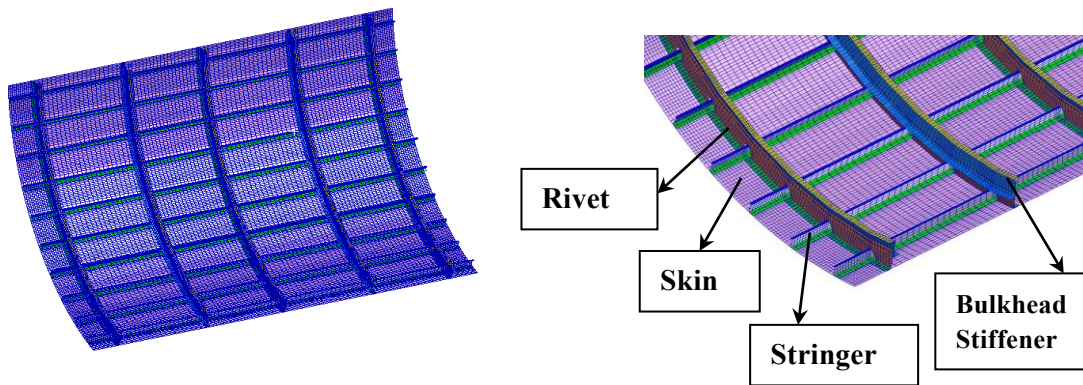


Figure 3. FE model of Curved Stiffened Panel

The stiffened panel was modeled in MSC Patran using two-dimensional elements like QUAD4 and TRIA3

elements and 1D beam elements. The bulkhead stiffeners and the longitudinal stringers were modeled by shell elements and are connected to the skin by rivets modeled using 1D beam elements. The design details are shown in Figure 4.

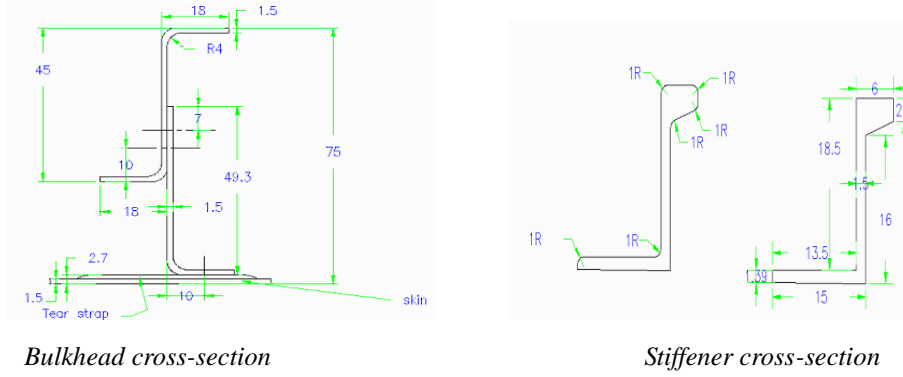


Figure 4. Stiffened panel modeling details

2.2.1 Loads and boundary conditions

The loads applied to the stiffened panel are taken from the global analysis of the fuselage structure at the boundary edges of the panel marked in the global model. Tensile load was applied in the hoop direction and each bulkhead was loaded with 1088 N and the skin of the panel was applied with 1760 N.

2.2.2 FE Analysis

A linear static analysis was performed for the stiffened panel with different crack lengths at the mid bay assuming the central stiffener to be broken and the stresses at the crack tip and the nodal displacements are noted for calculating the strain energy release rate, G and hence SIF by MVCCI method as shown below:

2.3 Calculation of stress intensity factor by MVCCI method

Stress Intensity factor,

$$K = G \cdot E \quad (1)$$

Strain energy release rate,

$$G = (1/2\Delta a) \cdot v \cdot f/t \quad (2)$$

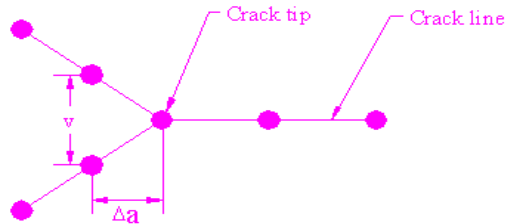


Figure 5. Illustration of MVCCI method

2.4 Residual strength calculation

Residual strength in the skin is calculated using

$$\sigma_{\text{residual}} = K_{IC} / K_a * \sigma_{\text{applied}} \quad (3)$$

Residual strength in the Stringer

$$\sigma_{\text{residual}} = (\sigma_{\text{applied}} * \sigma_{\text{ultimate}}) / \sigma_{\text{stiffener}} \quad (4)$$

Residual strength of rivet

$$\sigma_{\text{rivet}} = Q_{\text{max}} / A_{\text{rivet}} \quad (5)$$

Considering the shear strength of the rivet as 75% of the ultimate strength of the rivet material, we get the residual strength in the rivet as

$$\sigma_{\text{residual}} = (0.7 * \sigma_u * \sigma_a) / \sigma_{\text{rivet}} \quad (6)$$

3.0 RESULTS AND DISCUSSION

A linear static analysis of the entire fuselage was conducted and an overall structural response of the fuselage was analyzed to determine the maximum tensile stress location at the top skin of the fuselage, which is the possible location of crack initiation. It was observed that the stresses were maximum at the fuselage top skin and at the corners of the main door and the exit doors. Since the present interest is the location of the maximum tensile stresses at the top skin, a detailed stress variation at the top skin is analyzed. The stress distribution in the top skin shows a maximum stress of 105.9MPa, in the skin between the bulkheads 15 and 16 of the fuselage and the stresses at the adjacent bulkheads shows a uniform stress distribution and hence a stiffened panel comprising of 4 bays was selected near the maximum tensile stress location of the fuselage, consisting of 5 bulkheads and 9 longitudinal stringers for evaluating its crack arresting capability. The stiffened panel so selected is analyzed for the estimation of stress intensity factor for different crack lengths assuming that the central stiffener is broken.

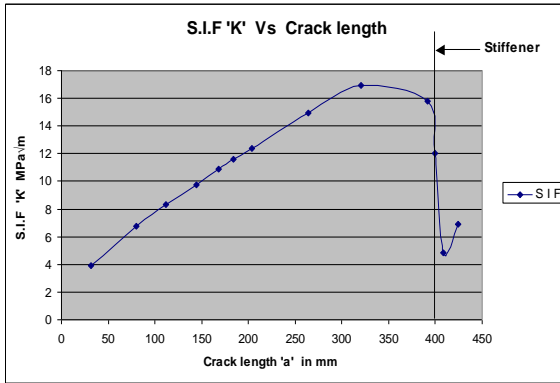


Figure 6. Graph of S.I.F Vs Crack length

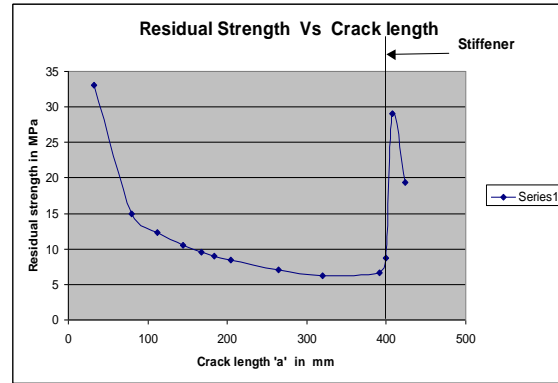


Figure 7. Residual strength of the skin

The values of SIF for the progressive crack lengths are obtained from the stiffened panel analysis by MVCCI method and are plotted to obtain the above characteristic curve shown in the Fig 5. It can be observed from the curve that the Stress intensity factor (SIF) of the panel varies linearly till it reaches the stiffener position up to the maximum SIF level of 17.55 MPa√m for the crack length of 320 mm. As the crack reaches the stiffener position the load is transferred to the stiffener through rivets, thus reducing the SIF near the stiffener, which ensures in

arresting the crack from propagating further. When the crack is propagated beyond the stiffener position, there will be an increase in SIF because the load shared by the skin increases gradually. The increasing trend in the curve is observed as the crack moves away from the stiffener position.

Residual strength of the skin is calculated for the progressive crack lengths and plotted as shown in the Figure 7. It can be observed that the residual strength of the skin decreases with the increase in crack length, indicated by the downward trend of the plot. As the crack reaches the stiffener position the load from the skin is transferred to the stiffener which reduces the residual strength of the skin is observed in the Fig 6. When the crack again propagates away from the stiffener, there is a decrease in the residual strength due to sharing of load by the skin is restored thus causing the curve to drop downwards.

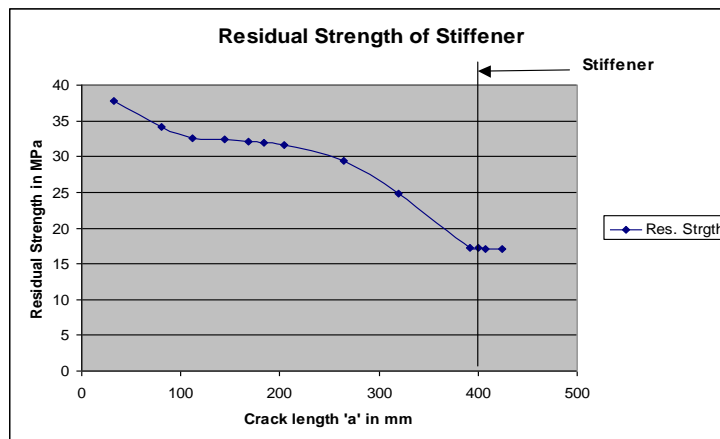


Figure 8. Residual strength of stiffener

Residual strength of the stiffener is calculated for the different progressive crack lengths and plotted in the graph shown in the Figure 8. It can be observed that the residual strength of the stringer decreases as the crack progresses towards the stiffener because of the increase in the load acting on the stringer due to transfer of load from skin to stringer and it can also be observed that the residual strength of the stringer is greater compared to that of the skin.

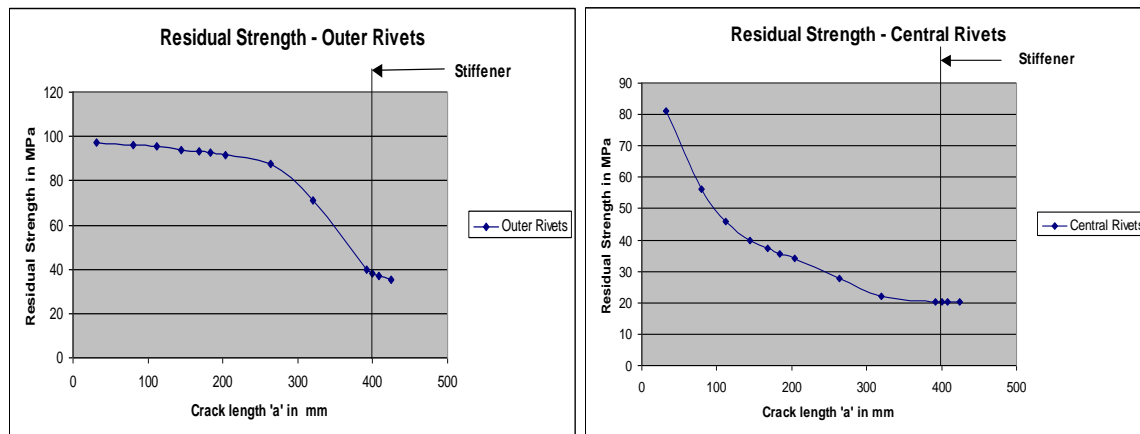


Figure 9. Residual strength of the outer rivets and the central rivets

The residual strength of the rivets is calculated for outer rivets away from the central stiffener and its variation with the increasing crack length is shown in the Figure 9. Initially residual strength of the outer rivets are high, because the crack is away from the outer stiffener and as the crack propagates near the stiffener region the residual strength of the rivet decreases due to load transfer from skin to the outer stiffener through the rivets. A plot of residual strength of the rivets of the central stiffener is also plotted for the progressive crack lengths shown in the Figure 9. Loads acting over the rivets of central stiffener is more than that of load acting over the rivets in the outer stiffener, this is because of the fact that the crack is assumed to propagate from the central stiffener towards outer

stiffener. As the crack length increases the residual strength of the rivets in the central stiffener decreases, which can be observed from the above plot.

3.1 Panel strength diagram

A combined plot of residual strength of all the panel elements is shown in the Figure 9 known as Panel strength diagram.

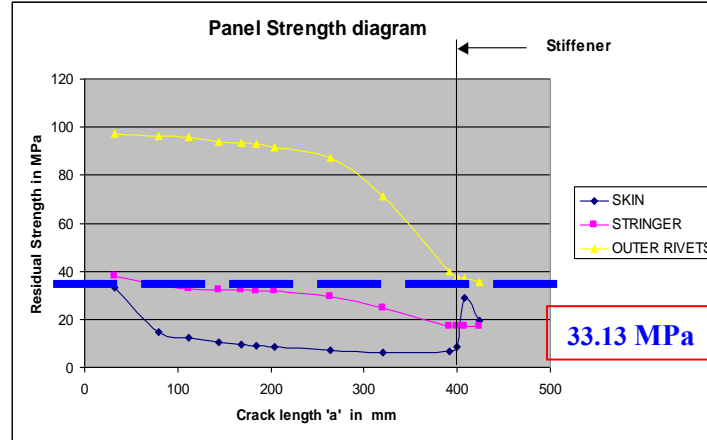


Figure10. Panel Strength diagram of curved stiffened panel

Here the residual strengths of cracked skin panel, the rivet and the outer stiffener are shown .In order to demonstrate a two bay crack arrest, one needs to show that none of the three break when the crack becomes 2-bay long which is clear from the above diagram. Also it is evident that if the failure occurs the skin will fail first, followed by the stringer and finally the rivets. If the rivets fail first, then the entire panel will fail without indication, resulting in catastrophic failure. It is observed from the panel strength diagram that the presently analyzed 2 bay long cracked stiffened panel can carry a remote stress of 33.33 MPa as indicated by the Figure 9.

4.0 CONCLUDING REMARKS

In this study, the entire fuselage of transport aircraft was modeled using MSC Patran and a linear static analysis was carried out under pressurization loads to pin-point the maximum tensile stress location in the top skin of the fuselage. The Stiffened panel for the present study was selected based on the maximum tensile stress location in the fuselage analysis was analyzed for its crack arresting capability by evaluating the stress intensity factor and residual strengths for various crack lengths. It is evident from the stiffened panel analysis that the residual strength of stiffened panel decreases with increase in the crack length. However as the crack length approaches stringer location the stringer will share the maximum load, resulting in higher residual strength in the skin leading to crack arrest. Residual strength of the stiffener is comparatively higher than the skin, and the residual strength of the rivets in the outer stiffener are higher than that of the broken central stiffener rivets. The residual strengths of the skin, stiffener and the rivets are individually estimated to generate panel strength diagram, which shows the crack arresting capability of the stiffened panel. It is observed from the panel strength diagram that the presently analyzed 2 bay long cracked stiffened panel can carry a remote stress of less than or equal to 33.33 MPa. Also, it is evident that if the failure occurs, the skin will fail first, followed by the stringer and finally the rivets.

ACKNOWLEDGMENTS

The author would like to acknowledge Mr.D.V.Venkatasubramanyam, Head, Structural Technologies Division, NAL and Dr.P.K.Dash Ex-Head, Structural Integrity Division, NAL, for the support and encouragement The author would place on record his gratitude to Dr.A.R.Upadhyya, Director, NAL for sponsorship and permission to publish this work

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APPENDIX I. NOTATION

G	Strain energy release rate
K	Stress intensity factor
K_{IC}	Fracture Toughness
R	Material resistance
E	Elastic modulus, Young's modulus
ν	Poisson's ratio
σ_{ult}	Ultimate strength
U_x, U_y, U_z	Global (Cartesian) x, y, z-displacement (DOF)
R_x, R_y, R_z	Rotation about global x-axis, y-axis, z-axis (DOF)
Δa	Incremental crack length
f	Force at the crack tip
t	Thickness of the crack tip element
v	Differential displacement
A	Cross sectional area
K_a	Stress intensity factor for the crack length 'a'
$\sigma_{applied}$	Applied stress
$\sigma_{residual}$	Residual stress
Q	Load